JPL Publication 96-26

Using Large Launchers for Small Satellites

Introductory Lecture to 10th Annual AIAA/Utah State University Conference on Small Satellites—September 16, 1996

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December 15, 1996

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Space Administration

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The research described in this publication was carried out by the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

ABSTRACT

The opportunities offered to piggybacks by commercial satcom launches should be exploited. In a number of cases, a mass margin is available for depositing satellites in geostationary transfer orbits (GTO).

An onboard propulsion capacity of 1 to 1.5 km/s opens the possibility for a small spacecraft in a GTO to achieve various missions in low Earth orbit or in planetary space.

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I. The Piggyback Offer

The main problem encountered by the promoters and builders of small satellites is to obtain a cheap launching. Piggybacking is the obvious solution. Opportunities have indeed been offered for secondary payloads. For many years the Delta and, more recently, the Taurus launch vehicles have offered slots at the top of the second stage. Users pay only the cost of integrating the payload in the launcher, typically \$2 to \$2.5 million. Unfortunately, the Delta and Taurus launches are heavily booked for the coming years.

Other options are open throughout the Russian space system. One already used is Cosmos SL-8, maybe the most successful rocket ever: 730 firings with a 97.4% success rate and a total of 670 satellites placed in orbit, including 40 simultaneous launches of 8 spacecraft. The current rate for the usual Russian customer is 4 to 6 per year launched at an 83-deg inclination and 1,000-km altitude; the total capacity is 30 launches per year. Launched from Plessetsk and Kapustin Yar, the Cosmos SL-8 can place 1,000 kg in a polar, 900-km circular orbit. It can accommodate secondary payloads up to 100 kg, launched at first opportunity with 3 months notice; examples are Astrid for the Swedish Space Corporation and Faisat for Final Analysis, Inc. It is now offering services through a joint venture called Cosmos USA, between Polyot Design Bureau of Omsk and Assured Space Access, Inc., of Arlington, VA.

Another option is Eurokot Launch Services GmbH founded in 1995 by Daimler Benz Aerospace of Germany and the Russian Khrunichev State Research and Production Space Center. It produces Rockot, a three-stage liquid-propellant launch vehicle composed of a former SS 19 strategic missile and a new upper stage called Breeze. Launched from Plessetsk (62.7°N, 40.3°E), a typical launch places an 830-kg payload in a 700-km altitude orbit at an inclination of 98 deg. The first commercial launch is now scheduled for the middle of 1998. A dual-launch concept is offered. The UoSat-12 experimental small satellite of 300 kg, which plans to test a water resistojet engine, is currently set up for launch at this time.

The main purpose of this publication is to discuss another type of piggyback, one that uses a geostationary transfer orbit (GTO)—a parking orbit in which a commercial satcom is placed before an apogee maneuver sends it to a geostationary orbit.

II. Ariane

The first launch of Ariane-I took place on December 24, 1979, from Kourou, French Guyana. Since then, Ariane, upgraded to the higher performing Ariane-IV, has proved to be a technical and commercial success. Recently, launch 91 took place, and only five failures have marred the record. Dedicated to the launching of geostationary satcoms, Ariane has captured 50% of the world's commercial space market. It was developed by public funding from Europe, with France paying 66% of the expenditure for Ariane-I to -IV and 45% for Ariane-V; since 1984, its services are sold by Arianespace, a private company.

Ariane-V is totally different from Ariane-IV, and for this reason should not bear the same name. The name was chosen by CNES to convince politicians that its development would be incremental. This incremental development costs 7 billion eurodollars (\$9 billion), but it has to be said that up to the first launch, the expenses were within the predicted budget. A qualified commercial system should be available by the end of 1997.

The basic kinship of Ariane-IV and Ariane-V resides in their mission: They are both commercial ventures, designed for dual launches of geostationary satcoms. We will take for granted that **in the next 10 years, geostationary satcoms will remain a major part of the commercial space market**, and therefore that **a significant mass in GTO will exist on a continuous and permanent basis**.

Arianespace offers three possibilities for piggybacking: the **ASAP platform** described below; the **Carrying Structure**, which can support piggybacks either with or without a second Main Passenger; and **a place as second Main Passenger** without the Carrying Structure but inside the regular platforms, called Mini-Spelda in the case of Ariane-IV or Sylda in the case of Ariane-V (Table 1).

Table 1. *Ariane auxiliary payloads (in kg)*

Vehicle	ASAP	Carrying Structure	Mini-Spelda (IV) or Sylda (V)
Ariane-IV	55	300 to 400	No limitations
Ariane-V	80 or 250	A few hundreds	No limitations

A. ASAP on Ariane-IV

Arianespace has developed a structure called ASAP (**A**riane **S**tructure for **A**uxiliary **P**ayloads) to carry and deploy small satellites for an Ariane mission dedicated to a Main Passenger. This kind of payload is referred to as an "Auxiliary Payload." The group of auxiliary payloads, not including the Main Passenger, carried during a launch is called "the aggregate" [1].

ASAP is a circular platform mounted externally to the interface between the vehicle equipment bay inner core and the main payload adapter. It provides installation for a spacecraft on an annular surface 420 mm wide and an internal diameter of 2,060 mm.

The maximum mass of an auxiliary payload, including adaptation to ASAP, must be less than or equal to 50 kg. The maximum total aggregate mass is 200 kg. The mass of the ASAP platform is 60 kg. The maximum number of auxiliary payloads is defined on a case-by-case basis. The maximum dimensions for the payload with its adapter to ASAP (to be provided by the customer) are 450 mm × 450 mm for the base and 450 mm in height. A greater height can be negotiated for particular flights. The price charged to the customer is $1 million.

An ASAP platform is not available on every launch: Its presence and the number of auxiliary payloads depend on the total mass of the Main Passenger. Usually one mission carries two Main Passengers.

Ariane-IV (Table 2) is, for the foreseeable future (1995 and beyond), launched every three weeks from the CSG (Centre Spatial de Guyane), at Kourou (7°N lat). However, use of an ASAP platform is not planned before the end of 1998 because of the heavy masses of the Main Passengers accepted on the manifest. Altogether, since one ASAP platform is not included if the number of passengers is less than two, a minimum 250-kg margin is needed.

Table 2. *Ariane-IV*

Piggyback Option	Volume	Separated Mass	Remarks
ASAP	As per ASAP User's Manual, Issue 2, Rev. 1: 450 mm × 450 mm × 450 mm. Larger dimensions can be accepted on a case-by-case basis depending on the main passenger geometry.	Up to 55 kg (5 kg more than that stated in the user's manual).	About \$1 M for ASAP (if shared, a minimum of \$0.6 M).
Carrying Structure	No user's manual. Studied on a case-by-case basis. Examples: (1) AMSAT: diameter, 2,300 mm; height, about 1,100 mm. (2) EQUATOR S: diameter, about 1,300 mm; height, about 1,000 mm. (3) ARSENE: diameter about 1,300 mm; height about 1,300 mm.	More than 55 kg. Implies an increase in the performance of the launch vehicle by adding strap-on boosters. Maximum typically up to 300 or 400 kg, depending on the mission (Delta performance with the launch vehicle version used by the Main Passenger).	About \$10 M.
Mini-Spelda	As per Ariane-IV User's Manual, Issue 1, Rev. 7.	No limitation other than that imposed by the adapter carrying capacity and launch vehicle performance. (Note: This volume can be used for a "composite" made of an auxiliary payload inside carrying structures.)	

A point worth pondering is the small number of auxiliary payloads really placed in orbit: Only 6 ASAP platforms have been used for 22 payloads up to August 1996. This is clearly due to the inadequacy of the GTO for space missions: The 11 Ariane launches of 1995 could have placed 44 50-kg spacecraft in orbit, instead of the 2 actually launched!

Ariane-IV will be phased out in 2000 or 2001.

B. ASAP on Ariane-V

On a standard geostationary mission, Ariane-V (Table 3) delivers the Main Passenger on a GTO with the following osculating parameters:

inclination, $i = 7$ deg

altitude of perigee, $z_p = 620$ km; altitude of apogee, $z_0 = 35{,}883$ km

argument of perigee, $\omega = 178$ deg; longitude of first ascending node, $\Omega \cong 10°W$

direction of Sun coincides with the major axis of the ellipse.

Table 3. *Ariane-V*

Piggyback Option	Volume	Separated Mass	Remarks
Ariane-V ASAP	600 mm × 600 mm × 600 mm	Up to 80 kg	Price TBD
Carrying Structure	No user's manual. Studied on a case-by-case basis. Example, AMSAT: diameter, 2,300 mm; height, about 1,100 mm. (Height could, probably, be increased by 200 to 400 mm.)	More than 80 kg. The maximum depends on the performance available. A few hundred kg can be launched in upper/lower position (dual launch) or in a single launch.	Price TBD
Mini-Sylda	As per Ariane-V User's Manual Issue 2, Rev. 0.	No limitation other than that imposed by the adapter carrying capacity and launch vehicle performance. Note: This volume can be used for a "composite" made of an auxiliary payload inside the carrying structures, or with the Ariane-V ASAP structure carrying up to three spacecraft weighing up to 300 kg each.	

The total mass available for spacecraft and adapter(s) is,

for single launch,	6,800 kg
for dual launch,	5,900 kg
for triple launch,	5,500 kg

The intermediate orbit (at the end of the H 155 cryogenic stage boost phase) has the following characteristics: apogee altitude, 1,300 km; perigee altitude, 50 km; inclination, 7 deg. Release of a payload might be possible on this orbit during ballistic flight.

Ariane-V can be used to reach low Earth orbit (LEO). The reference mission has the following parameters: inclination, 28 deg 30 min; altitude, 550 km; mass, 18,000 kg.

Two adapters are available for supporting the satellite during launch:

Speltra provides the largest useful internal diameter (4,570 mm); it will be available after the launch of V-503.

Sylda-V provides a smaller useful internal diameter (4,000 mm); its maximum height is 5,080 mm. It will not be available before the end of 1998.

The ASAP-V (Figure 1) is a flat circular platform as is the ASAP-IV. It is compatible with Speltra and **two** Main Passengers. It is also compatible with Sylda-5 and **one** Main Passenger.

On Speltra, ASAP-V can be inserted between the 2,624-mm plateau and the Main Passenger adapter; there it can hold a crown of eight 80-kg microsats (maximal dimensions: $h \times l \times w = 800 \times 600 \times 600$ mm) (Figure 2).

On Sylda-V, ASAP-V can be inserted in the space left vacant by the second Main Passenger in the case of a single launch (Figure 3). There it can accommodate four 300-kg minisatellites (h × diameter: $1{,}500 \times 1{,}300$ mm). In this configuration, different combinations are possible, for example, 2×300 kg + 4×80 kg.

The mass of the ASAP-V plateau is 180 kg for the microsatellites ($M < 80$ kg) and 250 kg for the minisatellites ($M < 300$ kg); a full mission can accommodate 1.5 metric tons of auxiliary payloads including ASAP-V. ASAP missions will be possible by mid-1998.

The status of an Auxiliary Payload is severely constrained by the safety requirements for the Main Passenger, which pays for the launch. The Auxiliary Passenger can be removed from the mission at any time. A mission analysis must prove the total absence of danger from the Auxiliary Payload. Arianespace charges \$1 million for installation, as it does for Ariane-IV; mandatory insurance costs \$10,000 for a spacecraft.

Since access to a GTO depends on the manifest of commercial launchers, it would be of great benefit to the space community if companies other than Arianespace were to offer a possibility of piggybacking. For instance, the Delta-III has a potential for auxiliary payloads that has not been transformed into a company policy for access to customers. If a common interface with auxiliary payloads on all the commercial launchers could be defined between the McDonnell Douglas Corporation, Arianespace, and other companies, a real opportunity would open to use a GTO on a regular basis.

Possible missions on a GTO have been listed [2]:

(1) *Small geosynchronous Earth-orbit (GEO) communications.* Currently, developing countries must lease transponders on large, expensive commercial satellites. With a small satellite in a GTO, the possibility exists for these countries to purchase their own small, dedicated satellite at a competitive price.

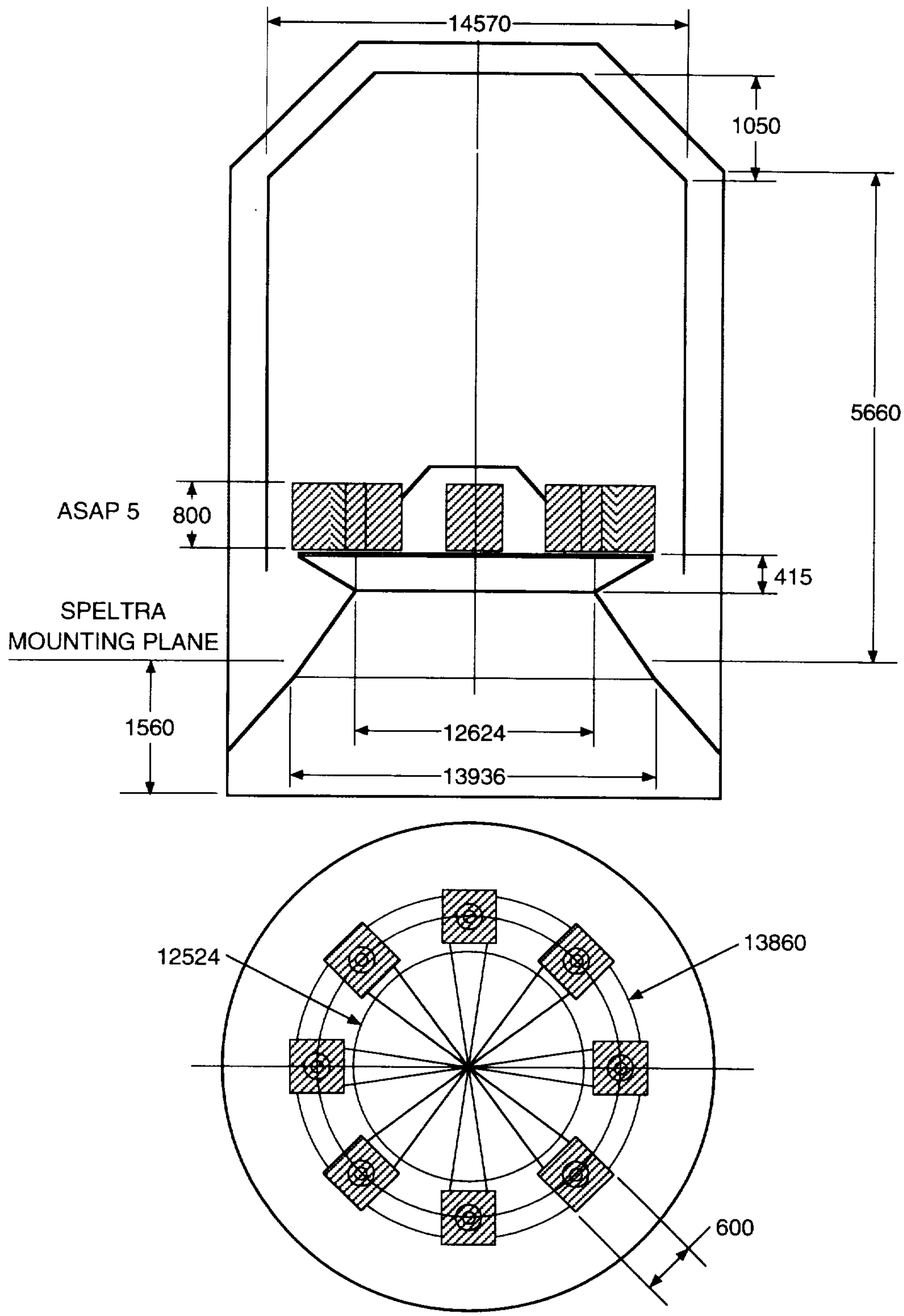

Figure 1. ASAP-V, configuration for eight microsatellites. (Valid for both Speltra and Sylda-V; dimensions in mm.)

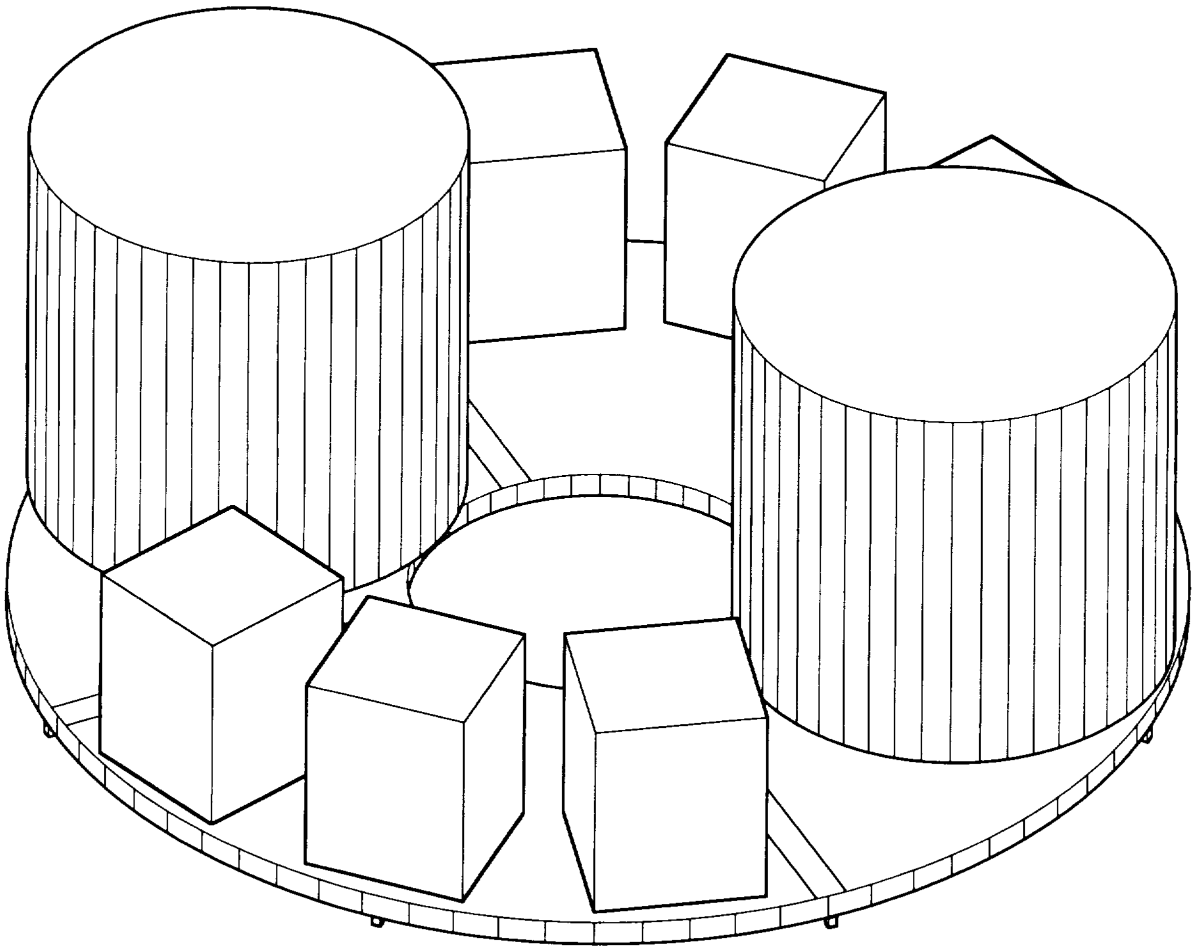

Figure 2. ASAP-V, configuration for two minisatellites and six microsatellites.
(Valid for Sylda-V.)

(2) *Meteorological monitoring*. Microsatellites have demonstrated their utility for localized weather monitoring from a LEO. Small satellites beginning in a GTO could be used as low-cost weather monitoring platforms, with the higher altitude providing more global access.

(3) *Geomagnetic data collection*. Because spacecraft in a GTO travel through the entire depth of the Van Allen radiation belts twice daily, they offer a unique vantage point from which to monitor such important phenomena in the space environment as solar wind and magnetic field interactions, galactic cosmic rays, and solar flares.

(4) *Ground-based astronomy calibration*. Ground-based optical astronomy is handicapped by the dynamic nature of Earth's atmosphere, which attenuates faint signals. A satellite in a very high Earth orbit with a low-power laser of known wavelength could provide the feedback necessary to perform real-time calibration and correction of these signals, greatly enhancing their resolution.

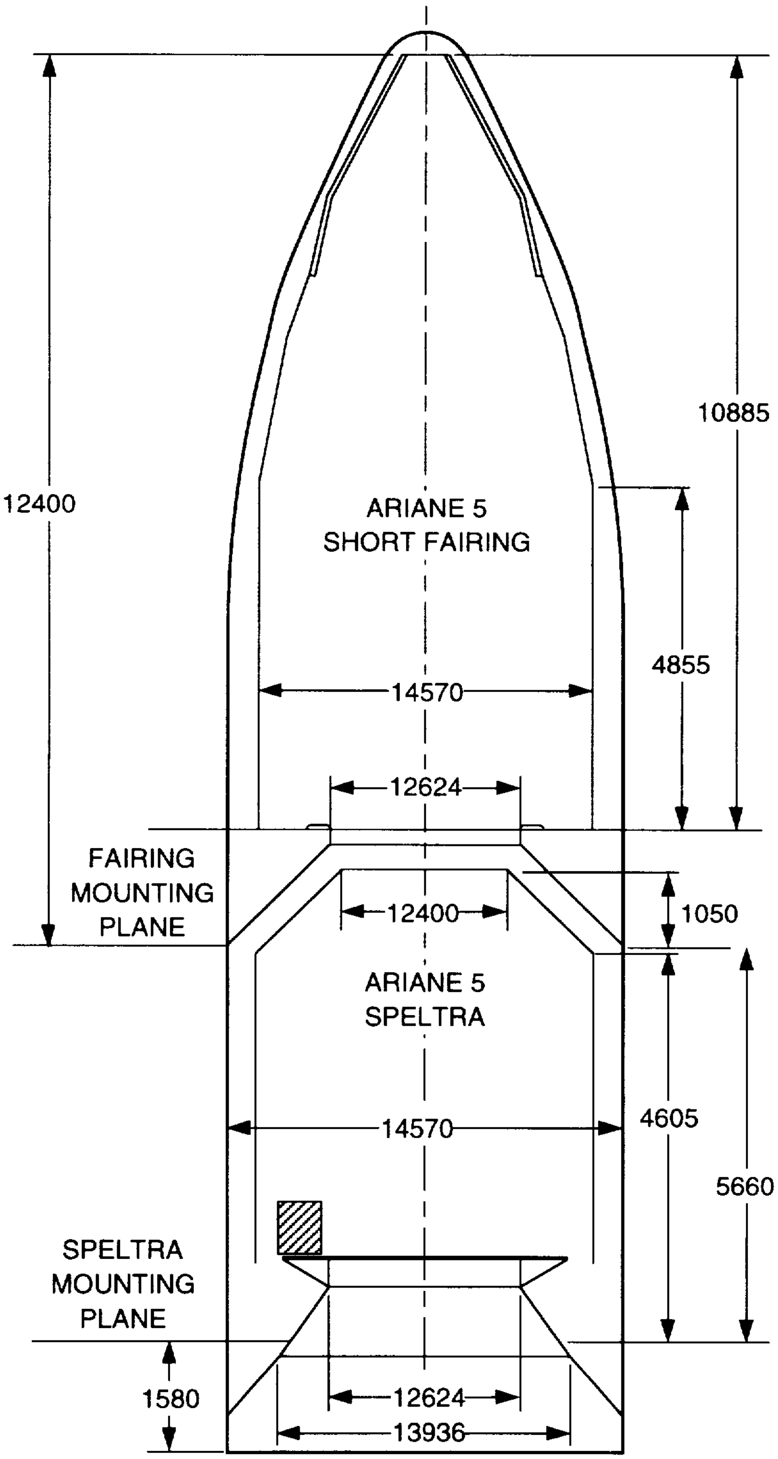

Figure 3. Position of a microsatellite in Ariane-V ASAP configuration. (Dimensions in mm.)

(5) *Lunar and planetary exploration.* From a GTO, the total velocity change (Δv) necessary to enter lunar orbit, visit Earth-approaching asteroids, or even other

planets is roughly equivalent to that needed for entry into a GEO. Spacecraft such as that used in the U.S. Clementine mission have demonstrated how very good planetary science can be conducted from small, relatively low-cost (~$70 million) platforms.

However, **it is a fact that the GTO opportunity has not been used: To make these missions happen, imagination is needed.**

III. Leaving a GTO

The lack of user interest in GTOs is well founded: These orbits are unstable (at least with a 200-km perigee, like that of Ariane-IV), they are dangerous because they traverse the radiation belts, and they are not adapted to many missions. This cheap access to space is, in fact, access to nowhere. This paradox has to be addressed.

The solution appears to me conceptually obvious: We must give our satellites a capacity for orbital maneuvers. I will briefly describe three types of such maneuvers: change of eccentricity, change of inclination, and escape from Earth.

A. Change of Eccentricity

It seems costly to go from a GTO to a 1,000-km circular orbit, but in reality it is cheap if we introduce the idea of **aerobraking**.

We start with our spacecraft in a GTO of 36,000 km apogee and 600 km perigee. By a velocity impulse Δv of about 50 m/s at apogee, we decrease the perigee to 130 km. At this altitude, atmospheric drag is large. After about 100 days and for an area of 3 m^2, the apogee has diminished to 1,000 km; the altitude of the perigee remains constant at 130 km. We now give a second apogee impulse of 104 to 234 m/s to increase the perigee to the value we want—between 500 and 1,000 km. The sum total of the Δv is in the range of 200 to 300 m/s (Table 4).

Table 4. *Aerobraking*

Maneuver	Location	Δv (m/s)	Effect
1	Apogee	50	Lowers perigee to 130 km
2	Apogee	150	Increases perigee to 675 km
Total:		200	

The technique has drawbacks:

(1) The spacecraft's 3-month stay in the radiation belts requires hardened components. A possible way out is to decrease immediately the apogee by a perigee maneuver of about 200 m/s to a value of around 25,000 km.

(2) Atmospheric drag can vary up to 50% from one day to another because of the solar-induced variability of atmospheric density: Careful control is needed at the end of the operations (called "walk out"). We believe that the satellite could execute these

operations autonomously by computing its orbit from GPS signals received on board.

(3) The concept has never been used on an Earth-orbiting satellite. However, as it has happened repeatedly, planetary exploration leads the way. Aerobraking has been used successfully by the NASA Magellan mission to Venus, and will next year constitute the baseline for the NASA mission Mars Global Surveyor that will orbit Mars. If it can be done on other planets, why not on Earth? CNES has chosen to experiment with this technique; it will be the basis for a joint mission with the Brazilian Space Agency: In 1999, a Franco-Brazilian 80-kg spacecraft will be transferred from an Ariane-V GTO to an 800-km circular orbit by aerobraking.

An obvious complement would be the use of a ballute—a balloon filled with low-pressure helium, inflated before first perigee, and trailed behind the spacecraft. The system will align itself with the drag and will not need attitude control. A sphere of 5 m radius inflated to a pressure of 3 mbars could be used at an apogee of 120 km to lose 100 m/s at each pass instead of 6 m/s, and the total maneuver could be accomplished in a small number of orbits. Thermal effects and g-loadings would be tolerable provided the ballute is made of kapton and held by a strong titanium wire and mesh. A mass of 10 to 12 kg is a reasonable estimate for the total ballute system.

The use of a ballute reduces the time of the operations dramatically. It should be tested, since the behavior of a ballute at a hypersonic velocity is unknown.

The method is extremely interesting when the apogee altitude is very high, as we will see when discussing the case of a change in orientation. However, it will presumably take a few years for this concept to be ready for mission applications.

Transforming aerobraking from acrobatics to a routine operation is an essential step since it could develop into one of the major assets in a modern strategy of access to space.

B. Change of Inclination

This maneuver also has the reputation of being expensive, and rightly so, but the requirements can be lowered by increasing the altitude of the apogee: The Δv impulse for rotation of the orbital plane is proportional to the velocity at apogee, which is itself inversely proportional to the altitude.

Starting with our spacecraft in a GTO,

(1) A velocity increase, Δv_1, at perigee raises the apogee.

(2) A Δv_i at apogee changes the inclination of the orbital plane.

(3) The altitude of apogee is decreased by aerobraking as before.

(4) Then the perigee altitude is increased, $\Delta v_p = 234$ m/s.

Two examples are given for different apogee values and 60 deg of rotation of the inclination (Table 5).

Table 5. *Change of inclination*

Maneuver	Location	Δv (m/s)		Effect
		Highest Apogee 71,000 km	Highest Apogee 340,000 km	
1	Perigee	337	683	Raises apogee
2	Apogee	909 (maneuvers 2 and 3)	210	Rotation to 60 deg inclination
3	Apogee			Lowers perigee to 130 km (Aerobraking)
4	Apogee	234	234	Increases perigee to 1,000 km (Aerobraking)
Total:		1,480	1,137	
Mass of propellant (kg)		160	130	

The mass of propellant indicated in Table 5 corresponds to a 400-kg spacecraft and a biliquid engine (I_{sp} ~ 295 s).

Here the difficulty lies in the time taken by the maneuvers, which is essentially the time taken by the aerobraking. Moving from the 340,000 km of apogee would take 240 days instead of the 90 days needed for 71,000 km of apogee. A solution would be the use of a ballute during the first passes at a low-altitude perigee, providing at once a large Δv, as seen before.

All figures presented here are conservative, and since a complete analysis has not been performed, we will conclude that a propulsion system with a Δv capacity of about 1 km/s gives access to practically any low Earth orbits starting from a GTO. The price paid is time. Table 6, due to C. Koppel (Société Européenne de Propulsion), provides the number of days to complete this maneuver for three propulsion options. The time required with plasma propulsion is proportional to the thrust, i.e., to the available electrical power. The following table has been established with an electrical power compatible with small satellites.

C. Escape From Earth

GTOs are ideal springboards for missions to interplanetary space. The spacecraft can stay in a parking GTO for a time, provided the components are hardened, and wait for the proper window. The only maneuver required for escape is an increase of velocity at perigee.

An interesting mission called **Blue Moon** is being pursued by the USAF Academy under the guidance of G. Moore and R. Humble [3]. For a Δv = 750 m/s, a 45-kg spacecraft is placed on an Earth–Moon transfer orbit timed to reach the Moon at the node of its orbit. This transfer—originally developed by Miller and Belbruno as a low-energy alternative to the Hohmann transfer—uses ballistic capture: The spacecraft at lunar arrival achieves an elliptic orbit about the Moon.

Table 6. *Time required for change of inclination*

Apogee (km)	Highest Type of Propulsion	Increase of Apogee (days)	Change of Inclination (days)	Aerobraking (days)	Circularization (days)	Total (days)	Mass of Propellant (kg)
340,000	Chemical	6	5	240 (30 with ballute)	3	254 44	125 for chemical
	Plasma plus chemical	6 (Chemical)	60 (Plasma)	240 (30 with ballute)	70 (Plasma)	376 166	10 for Xe 85 for chemical
	Plasma only	1,095	70	240 (30 with ballute)	80	1,485 1,275	49 for Xe
71,000	Chemical	3	5	90	3	101	155 for chemical
	Plasma plus chemical	3 (Chemical)	300 (Plasma)	90	75 (Plasma)	468	30 for Xe 44 for chemical
	Plasma only	511	330	90	80	1,011	49 for Xe

Gravitational effects of the Earth, Moon, and Sun are used to model the trajectory [4, 5, 6]. Ballistic capture can be obtained if the spacecraft goes into a region about the Moon where the dynamical effects due to the attraction of the Moon on the spacecraft and the perturbations of the Earth and Sun tend to balance, a region called the "lunar fuzzy boundary" (Figure 4) or "lunar weak-stability boundary." The spacecraft has to reach a distance from Earth of about 1.3 million km. At this distance, the effects are nonlinear, and the trajectory needs no Δv for a trip of about 3 months' duration. This type of transfer was used by the Japanese probe Muses A (Hiten) in 1990 [7]. Therefore, a Δv less than 1 km/s (total) provides the possibility of placing the spacecraft on a high eccentric lunar orbit; with a total Δv of 1,800 m/s, a circular orbit can be obtained. Blue Moon carries 11 kg of propellant, 7.4 kg of engine inert mass, and 21 kg of payload for a total of 45 kg (40% mass ratio, I_{sp} = 270 s).

The Belbruno technique for handling nonlinear dynamical systems could be extended to other missions. For instance, an escape from Earth's sphere of attraction could be obtained without Δv by jumping from one resonance situation to another in the Earth/Moon system. Cheap missions to asteroids and planets become possible. Belbruno and Marsden, in a paper to appear very soon, mention even the possibility of using the Jupiter fuzzy boundary for rendezvous with comets, in a fashion that makes a comet sample return easier [8]. This last mission, of course, could not be possible with a small satellite. Once more, the price paid is time.

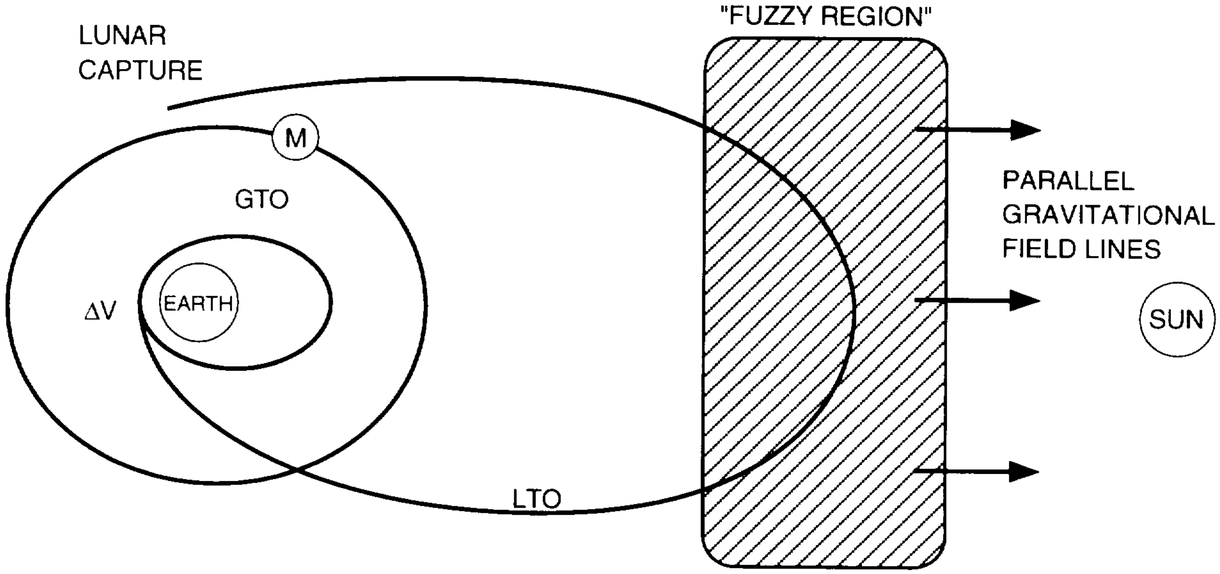

Figure 4. "Fuzzy boundaries," mission to the Moon and beyond.

If the fuzzy-boundary trajectories open an easy way from a GTO to the Moon, other techniques for which the temporal constraints can be accommodated are also available to reach the planets. P. Penzo [9] has shown that multiple lunar swing-bys can be used to reach Venus from Earth, in a mode similar to the technique proven in the mission to Comet Giacobini-Zinner [10]. A Belbruno trajectory followed by a double lunar swing-by 1 year later can send a spacecraft to Mars, but it seems that the time constraints may force the addition of a Δv maneuver at the last lunar pass. A simple solution has been found recently by P. Penzo who has shown that there exists a direct trajectory from a GTO to Mars at every opportunity for a Martian mission with a Δv of around 1.2 km/s [11]. The trick is to wait for several days to 2 months for the GTO lines of apsides, originally directed towards the Sun, to reach the direction of Mars. The trajectory is not much longer than the usual trajectories. The situation is the same for Venus, asteroids, and comets. Various options are available in matching the constrained GTO ellipse to the necessary escape direction toward the target body using lunar and Earth flybys.

From a GTO, then, a large number of orbits can be reached either around the Earth or in interplanetary space, provided a propulsion system with a capacity of 1 to 1.2 km/s be included. The mass of this system, including propellants, should not exceed one-third the total mass of the spacecraft. This can be achieved by the reduction in mass of all other components through miniaturization and the integration of the functions. The use of other new developments, such as positioning and attitude control by GPS for Earth orbits and efficient GaAs cascaded solar power generators, would increase the mass allocated to the payload.

IV. Orbit Control

Another reason for including a sizable propulsion capacity on small spacecraft is the necessity for orbit control in a LEO: The orbit control for satellites flying at low altitude is a problem of great importance for constellations.

Alenia Spazio [12] has investigated the feasibility of autonomous navigation combined with electric propulsion to achieve tight orbit control of small satellites injected at altitudes between 280 and 600 km.

The reference spacecraft has a frontal area of 1 m^2, a total mass of 300 kg, and a mean drag coefficient equal to 2.2. Using a standard GPS receiver, a positioning accuracy of about 100 m (1 sigma) has been considered.

The propulsion system includes a redundant ion engine with the thrusters parallel to the roll axis of the spacecraft body; its specific impulse is 3,000 s.

The results of this conceptual study show that long-term semiaxis variations, and in particular those due to drag, can be compensated automatically, keeping the maximum deviation from the desired orbiter to within a few meters of accuracy. Typical fuel (xenon) consumption is 5 kg/year to maintain an orbit altitude of 280 km (Table 7).

Table 7. *Fuel consumption for different propulsion systems*

Orbit Altitude (km)	Propulsion (hypothetical initial propulsion mass = 30 kg)				
	Consumption (kg/year)		Lifetime (year)	Mean Power (W) (a)	Mean Power (W) (b)
280 km	Chemical	48.3	0.6	6	
	Ionic	4.5	6.7	142	227
417 km	Chemical	6.1	5	6	
	Ionic	0.6	> 15	18	29
574 km	Chemical	0.9	> 15	6	
	Ionic	0.1	> 15	3	5

(a) Thrust duty = 100%

(b) Thrust duty = 60%

The concept will become reality when the Philips Laboratory spacecraft Mightysat, carrying a PPT (Pulsed Plasma Thruster), is deployed from the Space Shuttle in 1999 for a technical demonstration [13]. NASA Lewis Research Center, JPL, and Olin are involved in this development. With such a PPT (thrust of a few millinewtons, total mass of 6.6 kg, and power of 50 W), the lifetime of a small satellite jettisoned from the Shuttle's orbit can increase from a few days to nearly 2 years.

V. The Propulsion Systems

A. Introduction

Today, unfortunately, most small spacecraft lack one critical element that would allow full exploitation of the mission opportunities outlined above: **a propulsion system**. Until now there has been no need for very small, low-cost satellites to carry potentially costly instrument systems, since they were essentially experiments in technology. Over the years, various technical challenges in onboard data handling, low-power communications, autonomous operations, and low-cost engineering have been met and solved. Now, as mission planners look beyond passive missions in a LEO to the bold, new missions described above, the challenge of cost-effective propulsion has to be faced.

Obviously, all the capabilities needed to perform orbital maneuvering, orbit maintenance, and attitude control can be found in systems already used throughout the aerospace community. However, current off-the-shelf technology may not be appropriate for cost-effective applications within the context of small-satellite missions. Furthermore, the cost of systems procured using standard aerospace practices can be prohibitive. Thus, small-satellite mission planners face a dilemma: Future missions demand a propulsion capability that may be cost prohibitive, keeping the entire mission grounded.

Preliminary guidelines can be found in [2], which assesses system options. In this study, the chemical systems identified include traditional solid and liquid systems as well as hybrid systems. For electric systems, the study showed that resistojets and pulsed plasma thrusters (PPT) look the most promising for small satellites because of their low power requirement (50 to 500 W of continuous power). Ion systems have been designed for low power (~440 W) and long lifetime, but they are very expensive (~$1.5 million for each thruster) (Table 8).

Table 8. *Performance comparisons between various propulsion technologies analyzed*

System	I_{sp} (s)	Oxidizer/ Propellant Specific Gravity	Fuel Specific Gravity	Density $\times I_{sp}$ ($g\ cm^{-3}s$)	Thrust (N)	Power (W)
Bipropellant	290	1.447	0.8788	337.33	20	2
Hydrazine monopropellant	225	1.008		226.80	20	1
Hybrid	295	1.36	0.93	381.60	500	1
Cold-gas	65	0.23		14.95	0.1	0.5
H_20 resistojet	185	1.0		185.00	0.3	500
Solid (STAR 17-A)	286.7	1.661		476.21	16000	0
Hydrazine resistojet	304	1.008		306.43	0.33	500
Pulsed Plasma Thruster	1500	2.16		3240.00	7.0×10^{-4}	20
H_2O_2 monopropellant	150	1.36		204.00	1	1

B. Examples of Possible Propulsion Systems

1. Hybrid Rockets

These rockets are promising candidates for missions that require a Δv of 1 to 1.5 km/s. Providing an I_{sp} near 300 s, they offer an inherently safe option that uses a liquid oxidizer and a solid fuel; they cannot explode, they can use nontoxic chemicals, and they are cheap, but they still have to be tested in space.

The study performed by the University of Surrey [2] determined that a 200 m/s Δv motor would require a development cost of $100,000 and a total system cost of $170,000. The USAF Academy plans to use N_2O as a propellant and quotes a small cost.

Hybrid motors can be made restartable. Their gas can be used for a low-pressure attitude control system.

2. Plasma Thrusters

Used extensively for 25 years in the Soviet Union, these thrusters are excellent candidates for the station keeping of geostationary satellites. Delivering an I_{sp} of 1,500 s, they seem also to present a priori advantages for orbital maneuvers. As an example, for a 200-kg spacecraft in LEO requiring a Δv between 400 and 800 m/s, the utilization of the Russian engine called SPT 50 (25 mN, 400 W) would provide a gain of 50 kg compared with the use of a hydrazine engine [14, 15, 16]. Unfortunately, they also present drawbacks:

(1) They require a significant supply of energy, which prohibits the simultaneous use of payloads and thrusters.

(2) For some kinds of maneuvers, their thrust is weak and the maneuvers take a long time.

As an example of such very-thrust-sensitive maneuvers, the SPT 50 duration of the maneuvers described in Table 6 is over 3 years. With the SPT 100 (80 mN, 1,350 W), the delay is acceptable, but the electric power is not usually available.

3. Ion Engines

These engines provide a larger I_{sp} than the plasma thrusters, but they are heavier and require more power in the low-consumption domain. For instance, the JPL-developed N Star derivative "ultralight ion engine" has an I_{sp} of 3,300 s for 1 kW of peak power, but the total mass is close to 35 kg.

Among recent developments, LABEN is testing a xenon RMT, or Radiofrequency with Magnetic field Thruster (thrust 2 to 12 mN, power 70 to 420 W), which could provide to small satellites (100 to 1,000 kg) in LEO a Δv between 250 and 1,200 m/s. The problem of the lifetime of such small electric engines is open [17].

4. Chemical and Plasma Propulsion

The use of both chemical and plasma propulsion (25 mN) on the same mission provides marginal performance. The cost in propellant and xenon weight is not that small (90 kg) and the mission time is still of the order of 1 year (4.5 months with ballute). The system is complex and expensive.

5. Conclusion

Whatever the choice, which may not be easy for the time being, the study by the University of Surrey [2] shows that the necessary systems are indeed affordable (Tables 9 and 10).

The best systems available **today** remain classical chemical engines. For instance, for a price of the order of $1 million, the European company SEP offers a biliquid kit (MMH + N_2O) in the 200-N class, with an I_{sp} of 295 s, which has been used as a baseline for comparison with plasma thrusters in this presentation (Table 6). An engine of this type, of mass 300 kg (including 250 kg of propellant), could transfer a 500-kg spacecraft from a GTO to a circular 700-km orbit with a 65-deg inclination. These figures can be extrapolated linearly to a 300 kg spacecraft. In the USA, Rockwell Rocketdyne has developed an engine in the same class with similar performance, and its price is very low. Such an engine for the 400-kg class satellites or the hybrid engine for the 100-kg class satellites could provide access from a GTO to any Earth orbit within 100 days in the best of cases at a cost in weight of about 40% of the mass devoted to propulsion.

For very small satellites such as the OSC Picostar (12.5 kg), in a not-too-distant future, new chemical technologies are promising: the HAN based mono-ergol (which uses nontoxic ammonium nitrate), the cold gas using Tridyme™, electrolysis, and solid-fuel gas generation [18].

Table 9. *Summary of cost analysis results for traditional vs nontraditional commercial mission scenarios*

Mission	System	Propellant Mass (kg)	System Price ($)	Cost Figure of Merit
Traditional commercial	Pulsed Plasma Thruster	3.37	500,000	47.4
	Hydrazine resistojet	16.22	229,942	74.3
	H_2O resistojet	26.09	119,604	77.7
	Hybrid	16.69	171,701	84.3
	Bipropellant	16.97	176,987	84.9
	Hydrazine monopropellant	21.66	132,724	88.8
	H_2O_2 monopropellant	31.77	122,724	100.0
	Hydrazine resistojet	1.67	217,160	3.2
Nontraditional commercial	H_2O resistojet	26.09	119,604	56.2
	Pulsed Plasma Thruster	3.37	500,000	76.1
	H_2O_2 monopropellant	31.77	122,724	86.3
	Hydrazine resistojet	16.22	229,942	90.0
	Hybrid	16.69	171,701	91.7
	Hydrazine monopropellant	21.66	132,724	92.0
	Bipropellant	16.97	176,987	100.0

Table 10. *Summary of total system cost analysis results for an experimental mission and a lunar-orbit mission*

Mission	System	Propellant Mass (kg)	System Price ($)	Cost Figure of Merit
Experimental	Cold-gas	7.72	77,594	30.9
	H_2O resistojet	2.82	94,040	46.7
	H_2O_2 monopropellant	3.37	97,160	65.4
	Pulsed Plasma Thruster	0.34	200,000	67.4
	Hydrazine monopropellant	2.26	107,160	79.5
	Hybrid	1.72	171,701	89.2
	Hydrazine resistojet	1.67	217,160	97.9
	Bipropellant	1.75	176,987	100.0
Lunar orbit	Hybrid	106.18	248,393	49.0
	H_2O_2 monopropellant	165.72	237,762	59.1
	Hydrazine monopropellant	128.90	260,544	59.8
	Bipropellant	107.54	279,243	64.0
	Solid (2 × STAR 17-A)	108.46	1,420,000	68.5
	H_2O resistojet	148.98	272,988	69.6
	Hydrazine resistojet	103.80	332,198	100.0

VI. Conclusion

In the future, the development of new electric propulsion systems may lead to the use of transfer orbits different from the 600 to 36,000 km ellipse. The principles outlined in this paper will remain correct.

The concept of orbital maneuvers complements very well the concept of small satellites. More in-depth analysis could very well conclude that, with the help of GPS receivers for autonomous positioning and onboard orbit determination, orbital maneuvers that can be tested on small satellites could become an important method in the deployment of constellations of commercial or military importance.

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